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RESEARCH MEMORANDUM

MEASUREMENTS OF FLUCTUATING PRESSURES ON A $\frac{1}{4}$-SCALE MODEL OF THE X-1 AIRPLANE WITH A 10-PERCENT-THICK WING IN THE LANGLEY 16-FOOT TRANSONIC TUNNEL

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NATIONAL ADVISORY COMMITTEE
FOR AERONAUTICS

WASHINGTON

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SUMMARY

Pressure fluctuations have been measured near the trailing edge of the wing and near the leading edge of the tail of a $\frac{1}{4}$-scale model of the X-1 airplane with a 10-percent-thick wing in the Langley 16-foot transonic tunnel. The maximum values of pressure fluctuation measured at the wing and tail pressure-gage locations were about 0.6 and 1.6 times stream dynamic pressure, respectively. These values represent increases of more than tenfold in the pressure fluctuations as the angle of attack was increased at a constant Mach number. In the present investigation, no value of pressure-fluctuation coefficient could be used as a buffet criterion which would result in a buffet-boundary curve in complete agreement with the flight-determined buffet-boundary curve of the X-1 airplane. The particular vortex-generator installation used during part of the present investigation had no beneficial effects in reducing the amplitude of the pressure fluctuations at the pressure pickup locations on the wing or tail surfaces. Frequency analyses of the pressure fluctuations on the wing and tail indicate that, when the pressure fluctuations are large, they are random in frequency with pulsations noted at all frequencies from 10 to 1000 cycles per second, the frequency limits of the recording system.

INTRODUCTION

Buffeting is believed to be the response of an aircraft structure to aerodynamic-flow disturbances. Thus, the problem of obtaining the buffeting characteristics of specific aircraft from wind-tunnel tests presents many difficulties since both the structural and aerodynamic characteristics of the model and the aircraft are involved. Conventional

models for wind-tunnel tests are usually designed for general aerodynamic studies, and do not incorporate dynamic scaling of the structural characteristics of the airplane. An exploratory program has therefore been undertaken to determine whether the buffeting characteristics of airplanes can be predicted from a study of the aerodynamic flow disturbances on such models of airplanes. As part of this program, pressure fluctuations were measured on the wing and tail surfaces of a $\frac{1}{4}$-scale model of the X-1 airplane with a 10-percent-thick wing.

The pressure gages were located as far back on the wings as possible, as it was believed that pressure disturbances being shed by the wing would be indicated by gages in this location. In order to determine if the disturbances shed by the wing were influencing the flow at the tail, gages were installed near the leading edge of the tail directly downstream from the outermost wing gages.

Although it was realized that the information obtained from measurements made only near the wing trailing edge and the tail leading edge would be limited, it was hoped that the information obtained would correlate to some measure with the known buffeting characteristics of the full-size airplane, and would contribute to a basic understanding of the buffeting phenomena.

SYMBOLS

c	local wing chord parallel to plane of symmetry, ft
c'	mean aerodynamic chord of wing, $\frac{2}{S}\int_0^{b/2} c^2 dy$
S	wing area including area enclosed by fuselage (8.116 sq ft)
b	span (wing, 7.0 ft, tail, 2.85 ft)
y	spanwise distance outboard of plane of symmetry, ft
P_F	pressure-fluctuation coefficient, $\Delta p/q$
Δp	maximum peak-to-peak amplitude of the pressure variation across the diaphragm of the electrical pressure gage, lb/sq ft
q	dynamic pressure, $\frac{1}{2}\rho V^2$, lb/sq ft

ρ	density, slugs/cu ft
V	velocity, ft/sec
α	angle of attack of model fuselage center line, deg
M	Mach number
C_L	lift coefficient, L/qS
L	lift, lb

Subscript:

t	tail

INSTRUMENTATION AND TESTS

Model and Pressure Gages

The fluctuating-pressure measurements reported herein were made on an airplane model in the Langley 16-foot transonic tunnel. A detailed description of the tunnel, its operation, and calibration are presented in reference 1.

The basic model on which the fluctuating pressures were measured is a $\frac{1}{4}$-scale model of the X-1 airplane. The model has NACA 65-110 wing sections and modified NACA 65-008 tail sections and is the same model used in the investigation of reference 2. The incidence angle of the wing varies from $2\frac{1}{2}^\circ$ at the wing root to $1\frac{1}{2}^\circ$ at the wing tip. The incidence angle of the tail was 0° for all tests made during the present investigation. In addition to tests of the basic model, some tests were made with the horizontal tail of the model removed. General dimensions of the model are presented in figure 1, and photographs of the model mounted in the tunnel test section are presented in figure 2.

Part of the test program for the basic-model configuration was repeated with vortex generators installed on the wings. The vortex generators consisted of 0.125-inch by 0.125-inch flat plates approximately 0.015 inch thick with rounded leading and trailing edges. The plates were centered on the 27.5-percent-chord station, normal to the wing surface, and extended from the root to the tip of each wing. The included angle between adjacent plates was 15° and the distance between

the 50-percent-chord stations of adjacent plates was approximately 0.40 inch. Figure 3 illustrates the vortex-generator installation.

The wings and horizontal-tail surfaces of the model were equipped with a total of five NACA miniature electrical pressure gages of the type described in reference 3. Figure 1 illustrates the locations of the gages. The gages were installed in such a manner as to indicate the differential pressure between the upper and lower surfaces of the wing or tail in which they were mounted. As installed, the gages responded to pressure changes and with suitable auxiliary equipment, indicate the variation of pressure with time. The incidence angle of the wing is 2.2° at the spanwise station at which the outboard-pressure gage was installed. In addition to the gages installed on the model, a gage was installed in the tunnel wall so that fluctuating pressures on the model could be compared with fluctuating pressures measured at the tunnel wall. The gage installed in the tunnel wall was referenced to a relatively steady pressure through a long length of tubing which filtered out any pressure fluctuations present in the reference pressure.

Measurements and Reduction of Data

Pressure fluctuations.- Electrical signals proportional to the pressure pulsations experienced by each gage were amplified and simultaneously photographed on a recording oscillograph. Typical oscillograph records are shown in figure 4. The fluctuating pressures measured at the wing and tail gages of the model and at the tunnel wall have been converted to nondimensional coefficient form by dividing by free-stream dynamic pressure. The coefficients are designated as P_F. The values of fluctuating pressure used throughout this paper are the maximum peak-to-peak values obtained at each test point as illustrated in figure 4. Pulsations of large amplitude which occurred only occasionally were ignored. An examination of the oscillograph records obtained during the tests did not indicate that the natural frequency of the model on its cantilever support was a predominant frequency of the pressure pulsations.

Frequency analyses.- An arrangement was incorporated into the recording system so that the output of any gage could be switched from the recording oscillograph to a frequency analyzer. The analyzer and amplifier system as used for these tests had a lower frequency limit of 10 cycles per second and upper frequency limit of 1000 cycles per second. The equipment was usually operated over two frequency ranges: from 10 to 150 cycles per second and from 100 to 1000 cycles per second. The same constant width band-pass filter was used for both frequency ranges. The analyzer was calibrated to indicate root-mean-square values of the pressure fluctuations. The value indicated by the analyzer at any particular frequency is the average root-mean-square pressure fluctuation

occurring at that frequency over a period of time (estimated from considerations of the filter band width and sweeping times) of about 1 second for the low-frequency range and about 0.1 second for the high-frequency range.

Range of Tests

The Mach number range over which fluctuating pressures were measured extended from 0.70 to 1.00 and the model angle of attack was varied from about -4° to approximately 15°. As the test Mach number is increased, the maximum angle of attack at which tests can be made is decreased because of load limitations on the model support system. The Reynolds number for these tests varied from 4.1×10^6 to 4.7×10^6 as indicated in reference 2.

Accuracy of Measurements

Mach number and angle of attack.- The Mach numbers measured in the Langley 16-foot transonic tunnel are believed accurate to ± 0.005. The angles of attack presented are believed accurate to $\pm 0.05^\circ$.

Pressure-fluctuation coefficients.- The range of linear response of the pressure gages as used in the present investigation extends to approximately 2000 cycles per second. The range of linear response of the galvanometer elements of the oscillograph used for the present tests, however, extends only to about 500 cycles per second. As pressure pulsations were noted which contain frequencies up to at least 1000 cycles per second, the amplitudes indicated by the oscillograph for such pressure pulsations are too low. The errors due to nonlinearity of the galvanometer elements, reading of the records, and calibrations are such that the pressure-fluctuation coefficients presented in this paper are believed to be approximately 10 to 20 percent too low.

Frequency analyses.- The frequency scale on the figures presented in this paper are believed accurate to within ± 2 or 3 cycles per second on the low-frequency range and ± 20 or 30 cycles per second on the high-frequency range. Although the amplitude response of the frequency analyzer system as used in the present investigation is flat up to 1000 cycles per second, the indicated amplitudes are believed to be too large because of the response of the constant-width band-pass filter used in the equipment to signals containing random frequencies. The parameter being investigated with the frequency analyzer was predominant frequency rather than specific amplitudes at a given frequency. Thus no corrections have been applied to amplitudes of the pressure fluctuations indicated by the frequency analyzer. In order to determine if data could be

repeated with the frequency analyzer, two frequency analyses of the pressure fluctuations at the left tail gage were made within a period of about 3 minutes during which time the test conditions were held as close to constant as possible. These analyses are presented in figure 5. The ordinates on the frequency-analysis figures are root-mean-square values of the pressure fluctuations in pounds per square foot. Each small division on the ordinate scale represents 1 decibel. The largest differences in the frequency analyses shown in figure 5 occur between 15 and 20 cycles per second. Above 20 cycles per second, the differential pressure-fluctuation levels for the 2 records are approximately constant and equal although some differences occur in the locations of the small peaks.

RESULTS AND DISCUSSION

Fluctuating pressures have been measured on a $\frac{1}{4}$-scale model of the X-1 airplane with a 10-percent-thick wing in the Langley 16-foot transonic tunnel. In addition to measurements for the basic configuration, tests were made with the horizontal tail removed and for one vortex-generator installation. As the tests progressed, it was found that, in general, the three wing gages yielded approximately the same information while the results from the two tail gages were also similar to each other. Data are therefore presented only for the outboard pressure gage on the left wing and the pressure gage on the left tail.

Pressure Fluctuations

Although the gages were installed in the wing and tail of the model to indicate the difference in pressure between the upper and lower surfaces, it is believed that the pressure fluctuations presented in this paper for positive angles of attack are primarily pressure fluctuations on the upper surfaces of the wing and tail (see ref. 4).

Basic and tail-off configurations.- Plots of pressure-fluctuation coefficient P_F as a function of model angle of attack at various Mach numbers are presented in figure 6. Because of the wing incidence angle, the angle of attack of the wing spanwise station at which the outermost pressure gages are located is 2.2° greater than the model angle of attack.

Considerable scatter in the pressure-fluctuation coefficients is noted for some test conditions. It is believed that the scatter is caused by the unstable nature of the flow and is not inherent in the recording method or equipment.

In figure 6(a) the data obtained at the left-wing-outboard gage for the basic and tail-off configurations are plotted together as it was believed that the influence of the tail on the flow over the wing was slight. At a Mach number of 0.70 the pressure-fluctuation coefficients are relatively constant at a value of about 0.4 up to an angle of attack of about 6°. At an angle of attack of 7° the differential pressure fluctuations on the wing have increased abruptly. At an angle of attack slightly above 8° a peak value of differential-pressure-fluctuation coefficient of 0.54 occurs in the faired curve, and further increases in angle of attack result in a decrease in the differential-pressure-fluctuation coefficients until a value of about 0.17 is obtained at the maximum angle of attack of about 15°.

An examination of the chordwise static-pressure distributions obtained at a spanwise station near the outboard gage location on the left wing indicated that at a Mach number of 0.70 as the angle of attack is increased above 8°, there is a definite forward movement of the shock location and separation point at that spanwise station on the upper surface of the wing. The decrease in differential-pressure-fluctuation coefficient which occurs as the angle of attack is increased above about 8° at a Mach number of 0.70 is probably associated with this forward movement of the shock and separation point on the upper surface of the wing.

At Mach numbers of 0.80 and 0.85, compressibility effects become apparent and the rise in fluctuating differential pressure which occurs as the angle of attack is increased starts at a lower value of angle of attack than at a Mach number of 0.70 and is more gradual.

At a Mach number of 0.90, considerable difference is noted between the differential-pressure-fluctuation coefficients obtained for the basic and tail-off configurations particularly at angles of attack of about -2° and 0°. An examination of the static-pressure distributions measured for the two configurations at a Mach number of 0.90 indicates that, for the tail-off condition, the shock on the wing has moved forward from the position it assumed for the basic configuration. For the basic configuration, the shock is in the vicinity of the pressure gage location as evidenced by the static-pressure diagrams. If the shock moves across the electrical pressure gage, the differential pressure indicated by the gage changes abruptly. At $M = 0.90$ for $\alpha = -2.13°$ (see fig. 7) and for $\alpha = 0.17°$ the records from the oscillograph indicate that the shock was moving across the electrical pressure gage. For these conditions, two values of pressure-fluctuation coefficient are shown in figure 6(a). The flagged symbols represent the value of pressure-fluctuation coefficient obtained if the pressure variation due to the shock moving across the pressure gage is ignored (see fig. 7). The faired curve at a Mach number of 0.90 in figure 6(a) has been drawn considering the flagged

points. From a study of the static pressure distributions and the information presented in reference 4 the pressure fluctuation caused by shock oscillation would be expected to move forward on the airfoil as the Mach number is decreased.

At Mach numbers of 0.95 and 1.00 the shock is behind the pressure-gage location and the pressure fluctuations remain practically constant at a relatively low value at all angles of attack investigated.

Plots similar to those in figure 6(a) for the outboard pressure gage on the left wing are presented in figure 6(b) for the left-tail gage and the tunnel-wall gage. At a Mach number of 0.70 the pressure-fluctuation coefficients at the left-tail gage are essentially constant at a relatively low value over an angle-of-attack range from -4° to 3°. At an angle of attack of 7° the pressure fluctuations at the left-tail gage have increased abruptly as they did at the left-wing gage. As the angle of attack is increased from 7° to about 10° the pressure fluctuations do not change abruptly but a gradual increase in the pressure fluctuations begins at 9°, and a value of P_F of 1.35 is noted at about 13° angle of attack. At angles of attack of about 14° and 15° the pressure-fluctuation coefficients were larger than 1.35, but because of improper adjustment of the equipment at these angles of attack the extremely large and rapid fluctuations in pressure did not leave a readable trace on the photographic record.

It is believed that the sharp rise which occurs in P_F at the left-tail gage at an angle of attack of 7° is not due to abrupt changes in the aerodynamic characteristics of the tail, but to the influence of the wing which exhibits a sharp rise in P_F at the pressure-gage location at an angle of attack of 7°. The tail section is 2 percent thinner than the wing section and is operating at the model angle of attack which is 2.2° less than the angle of attack of the wing spanwise station at which the outermost pressure gages are located.

At angles of attack below 7° at a Mach number of 0.70, the pressure fluctuations at the tail pressure gage are considerably larger than those measured at the wing pressure gage probably because the tail gage, being located near the tail leading edge, is more sensitive than the wing gage to angular variations in the flow.

As the Mach number is increased to 0.80, 0.85, and 0.90 the increase which occurs in the pressure-fluctuation coefficient at the left-tail gage as the angle of attack is increased becomes more and more gradual, and as was found with the wing pressure gage, does not occur at all at Mach numbers of 0.95 and 1.00 for the angle-of-attack range investigated. This result is in agreement with the results presented in reference 4 for two-dimensional airfoil tests.

If the present tests could have been extended to higher angles of attack than were attained at Mach numbers of 0.95 and 1.00, the pressure fluctuations at the pressure-gage locations on the wing and tail probably would have increased at these higher angles of attack from the values shown in figure 6. Note in figure 6(b) that although the differential pressure fluctuations measured at the left tail of the model increase by a factor of approximately 10 as the model is varied through the angle-of-attack range at a Mach number of 0.70, the pressure fluctuations measured at the tunnel wall indicate only a slight gradual rise as the angle of attack is increased. At higher Mach numbers than 0.70 the pressure fluctuations at the tunnel wall remain essentially constant at relatively low values as the angle of attack of the model is varied. Thus it may be assumed that the marked changes in pressure-fluctuation coefficient on the model are aerodynamic effects for this particular model. Measurements made in the center of the stream of the test section with an electrical pressure gage installed to indicate the difference in pressure between diametrically opposite points on the surfaces of a 3° cone indicated pressure fluctuations approximately one fifth of those measured at the tunnel wall.

In some instance the pressure fluctuations measured at the outboard pressure gage in the left wing are lower than the pressure fluctuations measured at the tunnel wall, probably because the pressure gages were installed in the wing and tail of the model to measure the difference in pressure between the upper and lower surfaces. Pressure fluctuations which are in phase with each other on the upper and lower surfaces would thus tend to cancel. This cancelling effect would not be obtained with the gage in the tunnel wall.

As previously mentioned, the increase which occurs in the pressure-fluctuation coefficients at the wing and tail-gage locations as the angle of attack is increased is more gradual at a Mach number of 0.80 than at a Mach number of 0.70. This finding is in agreement with unpublished data which indicates that for the X-1 airplane the onset of buffeting in the shock region is gradual compared to that of the stall region. Buffeting is considered to change from the stall region to the shock region at that point where the buffet boundary curve no longer coincides with the maximum lift curve. This point occurs at a Mach number of approximately 0.72 for the X-1 airplane. Thus, at a Mach number of 0.7 the onset of buffeting of the airplane is abrupt. For the $\frac{1}{4}$-scale model, the increase in pressure-fluctuation coefficient is also abrupt. At a Mach number of 0.8, the onset of buffeting of the airplane is more gradual than at a Mach number of 0.7 and the increase in differential-pressure-fluctuation coefficient for the model is more gradual than at a Mach number of 0.7.

In figure 8 the pressure-fluctuation-coefficient intensities are presented in such a manner that they can be compared with the flight-determined buffet boundary of the X-1 airplane. The lift-coefficient data required to prepare figure 8 are presented in figure 9. The flight-determined buffet boundary represents a variation in airplane normal-force coefficient of 0.01. Pressure-fluctuation-coefficient intensities of 0.05, 0.10, and 0.20 are shown for the left-wing gage in figure 8(a) and 0.15, 0.20, and 0.25 for the left-tail gage in figure 8(b). The data shown in figure 8(a) at a Mach number of 0.90 was based on the faired curve of figure 6(a) at a Mach number of 0.90, and the pressure fluctuations caused by the shock moving across the pressure gage have not been included.

Although the correlation between the flight-determined buffet boundary and a differential-pressure-fluctuation-coefficient intensity of 0.10 for the left-wing-outboard gage is relatively good over a small Mach number range, no value of differential-pressure-fluctuation coefficient exists either for the wing or tail gage which could be used as a buffet criterion to establish a buffet boundary which would completely agree with the flight-determined buffet boundary. The probable reason for this is the lack of a suitable number of pressure gages. A pressure gage fixed at one location on the wing or tail surface is influenced by local aerodynamic effects which may only slightly influence the buffeting of the airplane. For example, the results shown in reference 4 for rigid airfoils in a two-dimensional stream indicate that when the amplitude of the pressure fluctuations at some point on the airfoil is relatively large the amplitude of the pressure fluctuations varies considerably over the chord of the airfoil.

Vortex generators.- In figure 10 the data obtained with the vortex generators installed on the wings are compared with the curves which were faired through the data obtained for the basic and tail-off configurations. At a Mach number of 0.90, the shock wave on the wing is located in the vicinity of the pressure gage and, as for the basic configuration at a Mach number of 0.90, two values of P_F are noted for the points at negative angles of attack. Generally, the data obtained with the vortex generators installed are in approximate agreement with the data obtained for the basic configuration. In some instances, however, (left-tail gage at $M = 0.80$ and $M = 0.90$) the data obtained from the pressure gages with the vortex generators installed indicate that the buffeting characteristics would be expected to be inferior to those of the basic configuration if the data obtained at the gage location of the left tail is typical of conditions existing elsewhere on the tail. It should be remembered that in the present investigation only one vortex-generator configuration which may not have been an optimum configuration was considered.

Frequency Analyses

Typical records from the frequency analyzer are shown for the basic configuration in figures 11 to 13. Two frequency ranges were investigated: from 10 to 150 cycles per second and from 100 to 1000 cycles per second.

In figure 11 the frequency analyses are shown for the left-wing-outboard gage at a Mach number of 0.70. At an angle of attack of 4.9° the only disturbances which appear from 10 to 1000 cycles per second occur below 20 cycles per second, and are small. At an angle of attack of 7° for which the sharp increase in pressure fluctuation occurs (fig. 6(a)), pressure fluctuations are noted over the entire frequency spectrum for which data were obtained and no particular frequency could be considered to predominate.

In figure 12 the frequency analyses are presented for the left-tail gage at a Mach number of 0.70. At an angle of attack of 4.9° (fig. 12(a)), predominant frequencies are noted at about 10 cycles per second, and between 40 and 50 cycles per second. From approximately 70 to 1000 cycles per second, no pressure disturbances are noted above 2 pounds per square foot. At an angle of attack of 7.0° (fig. 12(b)), however, frequency characteristics similar to those obtained for the wing gage at a corresponding angle of attack are noted. Pressure fluctuations occur at all frequencies from 10 to 1000 cycles per second, indicating that the flow fluctuations are random in frequency. Additional records from the frequency analyzer have been included as figure 13 to indicate that the random-type flow fluctuations also exist for lower values of differential-pressure-fluctuation coefficient than the values for which it was shown to occur in the previous figures. At a Mach number of 0.8 and an angle of attack of -2.0° (fig. 13(a)), the principle pressure fluctuations occur in two frequency bands: from about 40 to 100 cycles per second and from approximately 125 to 150 cycles per second. Above about 150 cycles per second, no pressure pulsations above 0.7 pound per square foot were noted. When the angle of attack is increased to 2.8° at a Mach number of 0.8 (fig. 13(b)), it is again noted that practically all frequencies from 10 to 1000 cycles per second are represented.

The random-type flow experienced over the wing and tail of the model indicates that the aircraft designer has little chance of designing component parts of an aircraft with natural frequencies completely above or below those likely to be encountered in flight.

CONCLUSIONS

From a study of the pressure fluctuations measured near the trailing edge of the wing and the leading edge of the tail of a $\frac{1}{4}$-scale model of

the X-1 airplane with a 10-percent-thick wing in the Langley 16-foot transonic tunnel, the following conclusions may be made:

1. The maximum values of pressure fluctuation measured at the wing and tail pressure-gage locations were about 0.6 and 1.6 times stream dynamic pressure, respectively. These values represent increases of more than tenfold in the pressure fluctuations as the angle of attack was increased at a constant Mach number.

2. In the present investigation, no value of differential-pressure-fluctuation coefficient could be used as a buffet criterion, either for the tail gage or the wing gage, which would result in a buffet-boundary curve in complete agreement with the flight-determined buffet-boundary curve.

3. The vortex generators used on the wings of this model had no beneficial effect in reducing the amplitude of the differential pressure fluctuations at the location of the pressure pickups on the wing or tail.

4. Frequency analyses of the differential pressure-fluctuations on the model for conditions where maximum pressure fluctuations were found to be large indicate that usually the differential pressure fluctuations are random in frequency with pulsations noted at all frequencies from 10 to 1000 cycles per second, which are the frequency limits of the recording equipment.

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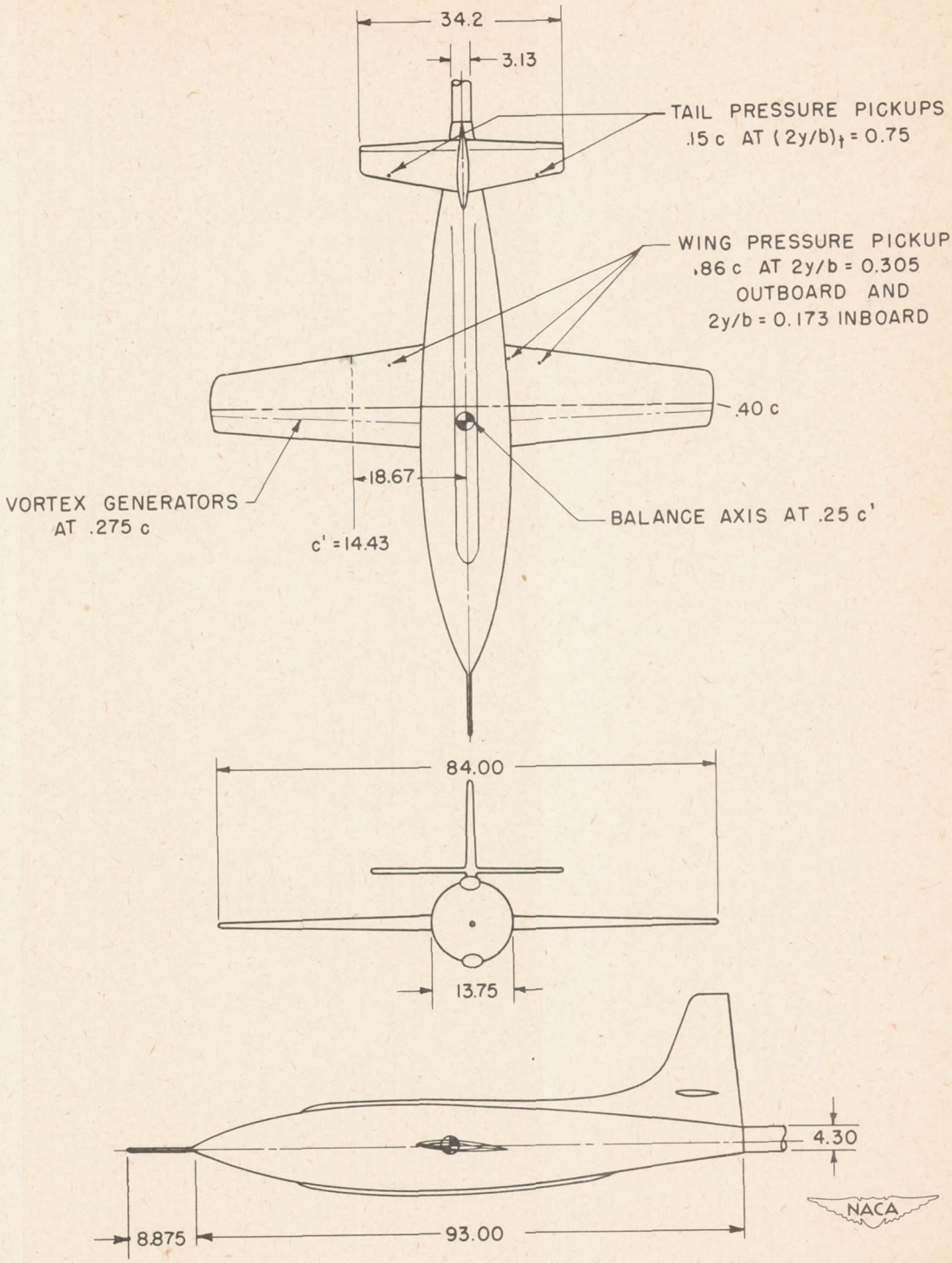
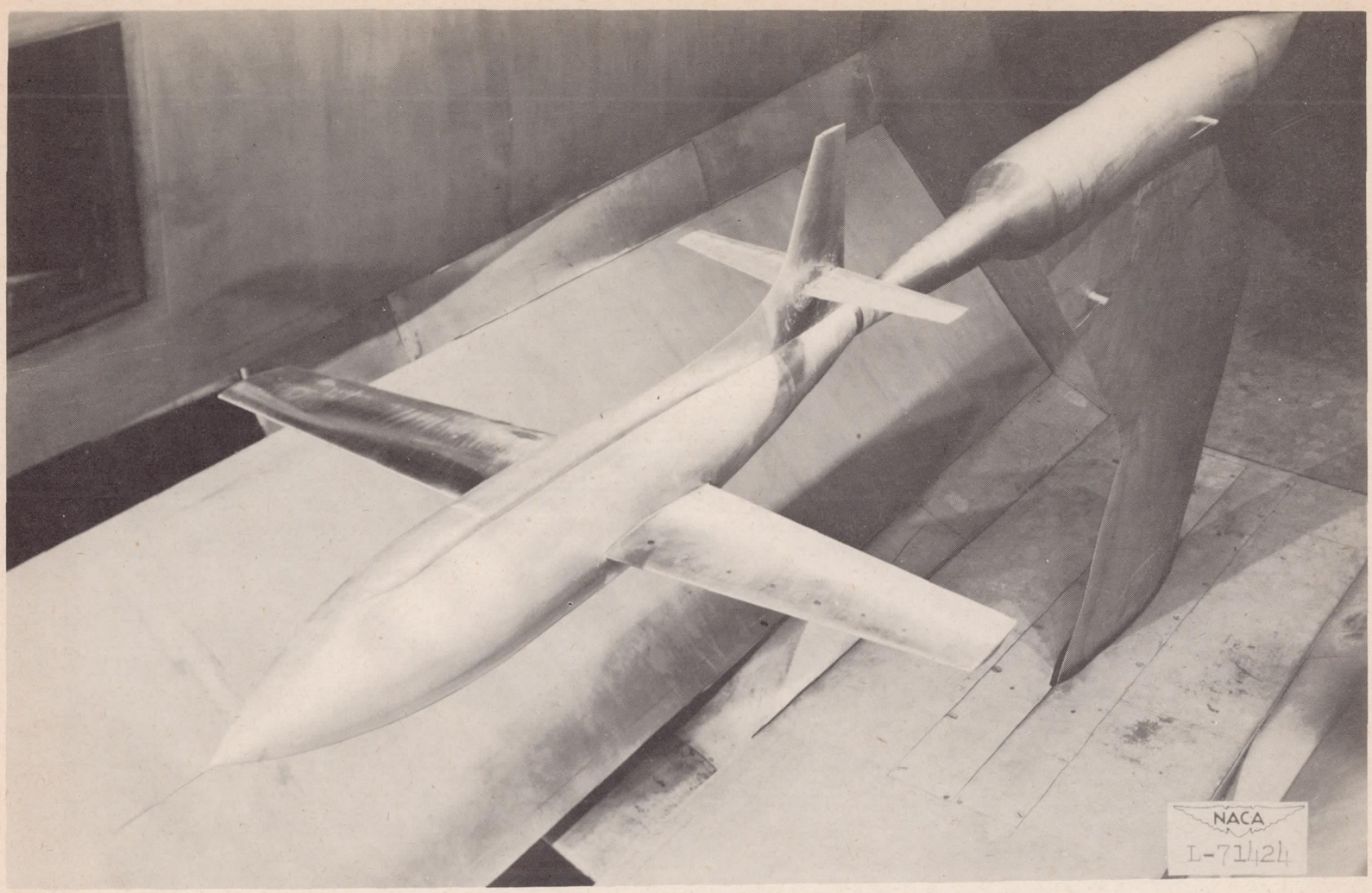
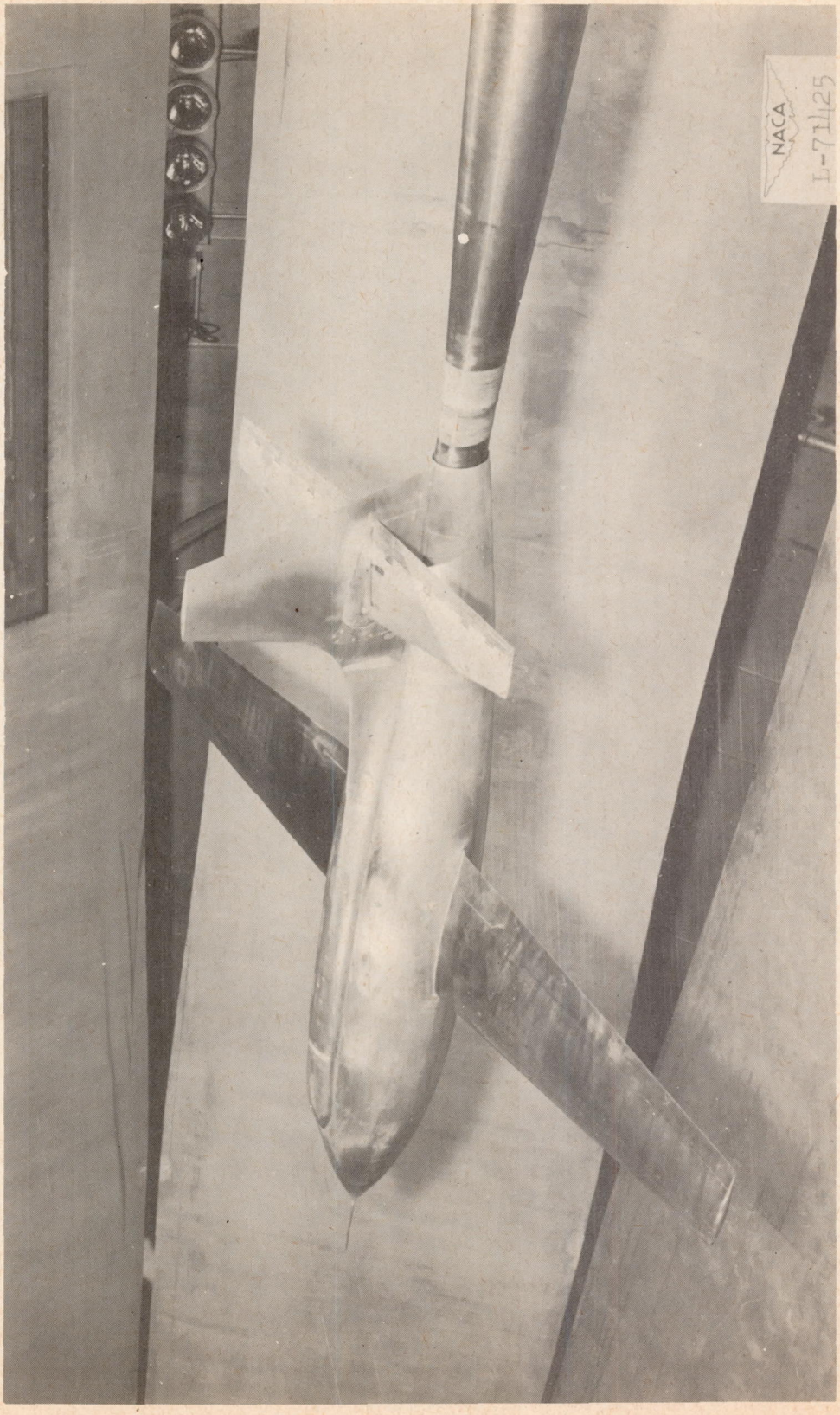


Figure 1.- Sketch of $\frac{1}{4}$- scale model of the X-1 airplane as tested in the Langley 16-foot transonic tunnel. All dimensions are in inches.



(a) Three-quarter front view.

Figure 2.- One-quarter-scale model of the X-1 airplane in the Langley 16-foot transonic tunnel.



(b) Three-quarter rear view.

Figure 2.- Concluded.

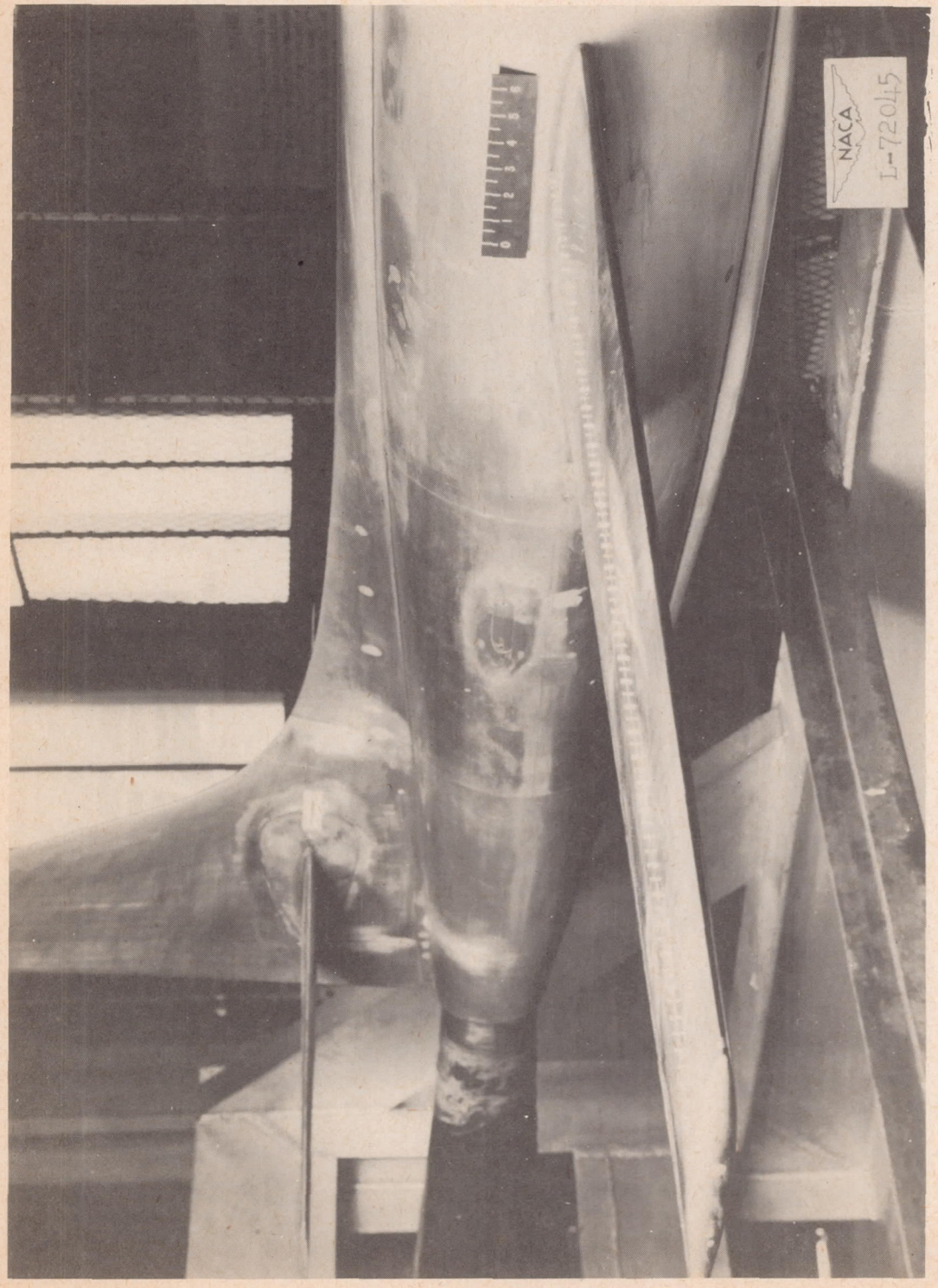
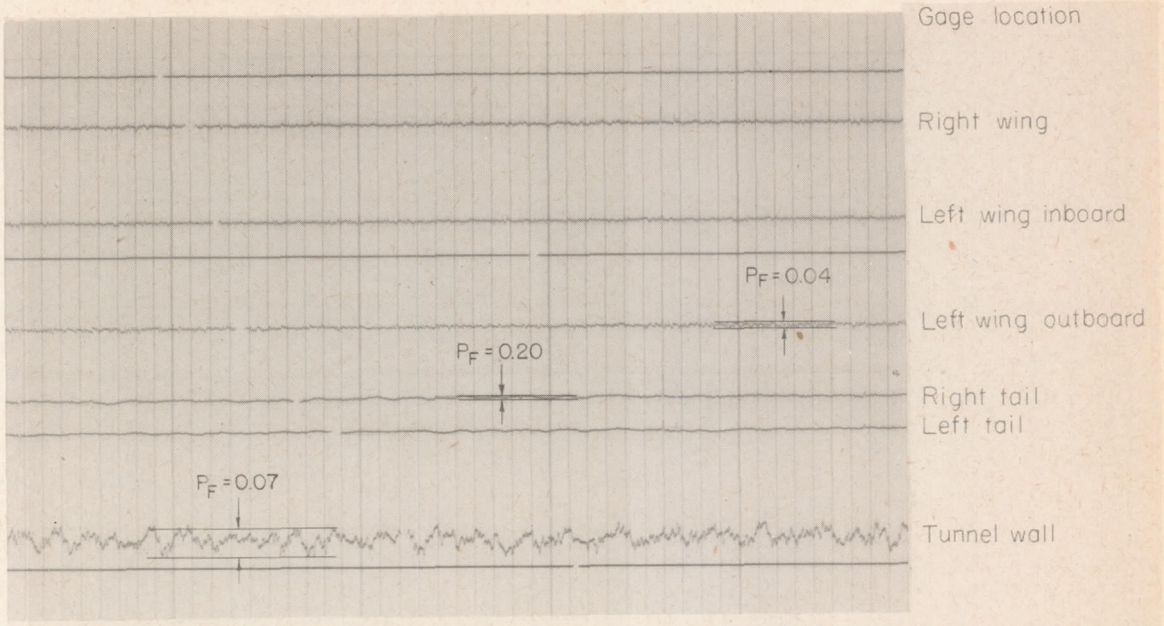
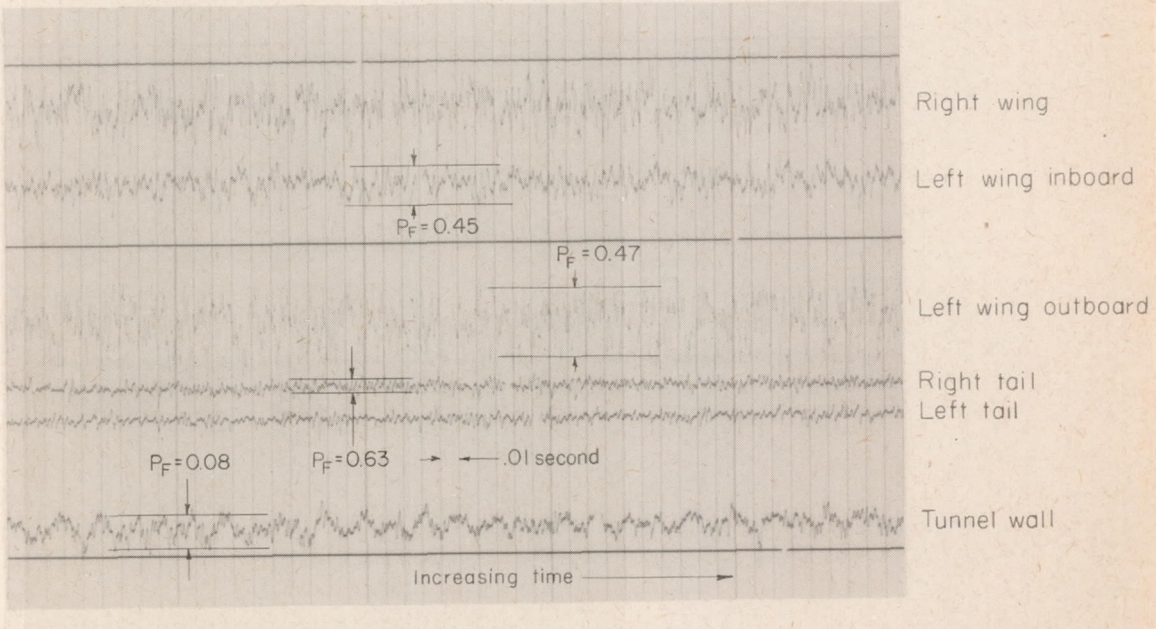


Figure 3.- Vortex generators installed on the model wings.



(a) Model angle of attack, α = 4.9°.



(b) Model angle of attack, α = 7.0°.

Figure 4.- Portions of typical oscillograph records. Mach number, 0.70.

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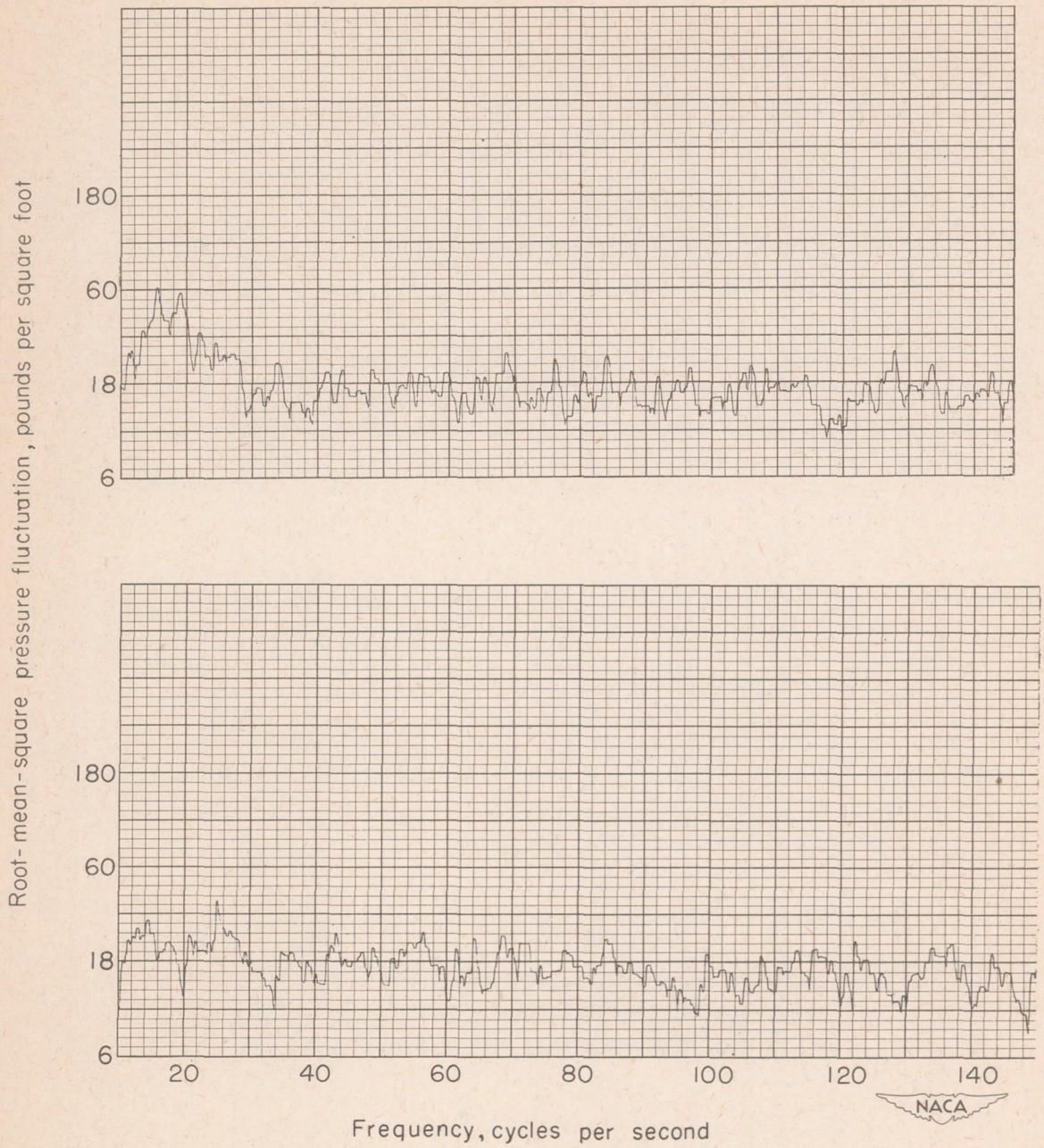
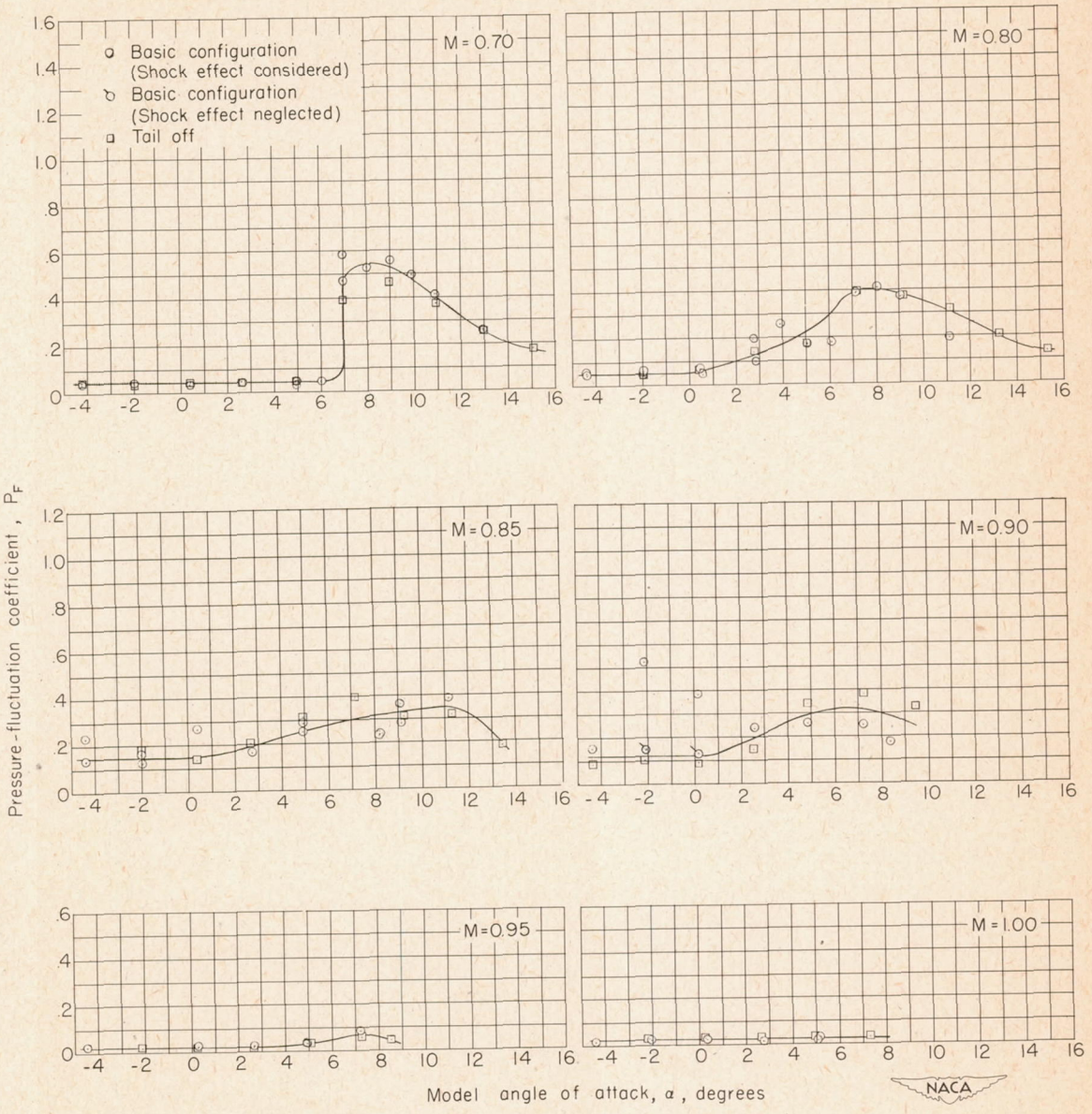
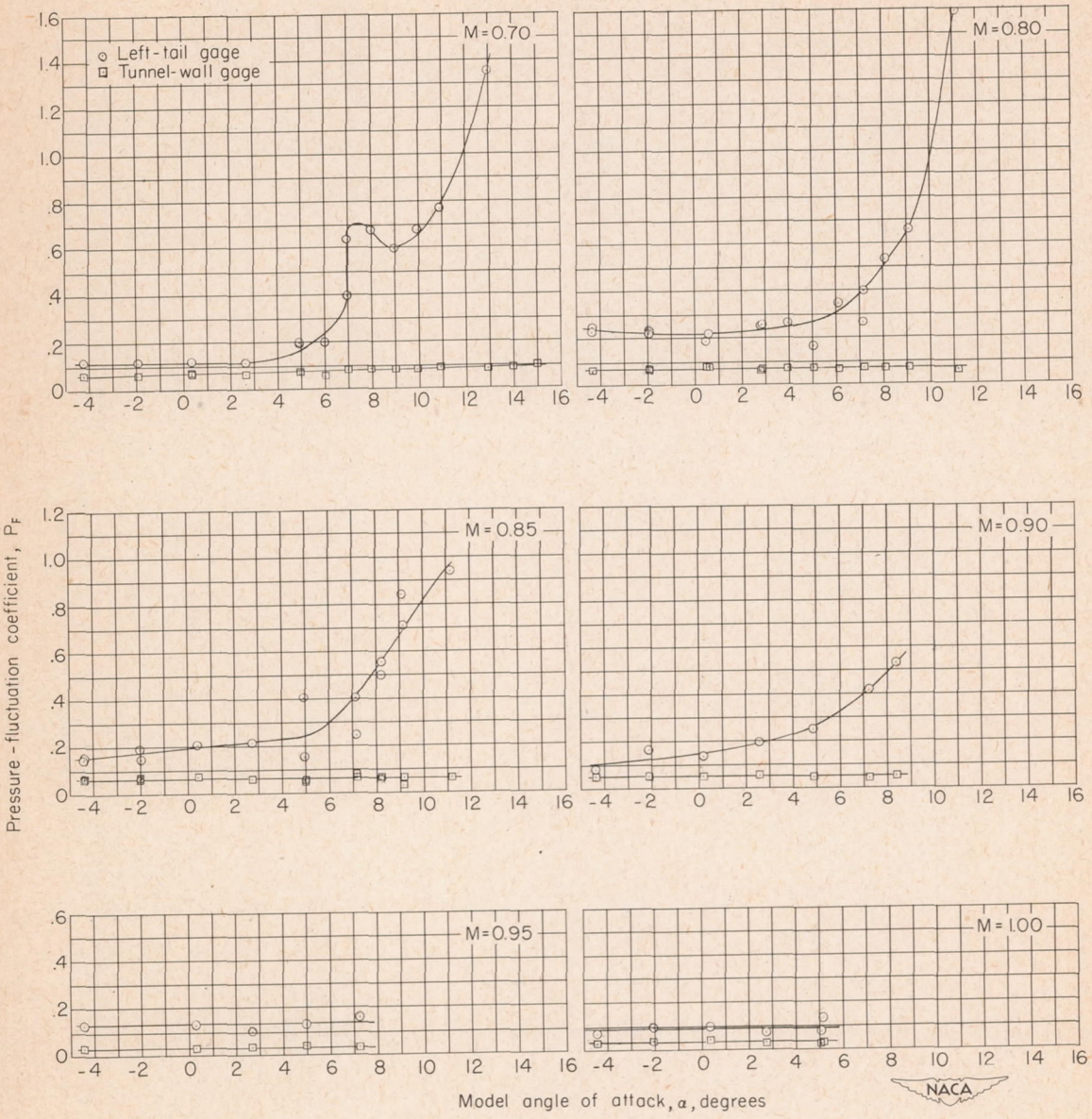


Figure 5.- Comparison of frequency analyses obtained for identical test conditions. Left-tail gage. $M = 0.74$, $\alpha = 7.0^{\circ}$.



(a) Left-wing-outboard gage Basic and tail-off configuration.

Figure 6.- Variation of pressure-fluctuation coefficient with model angle of attack at constant Mach numbers.



(b) Left-tail gage, Tunnel-wall gage. Basic configuration.

Figure 6.- Concluded.

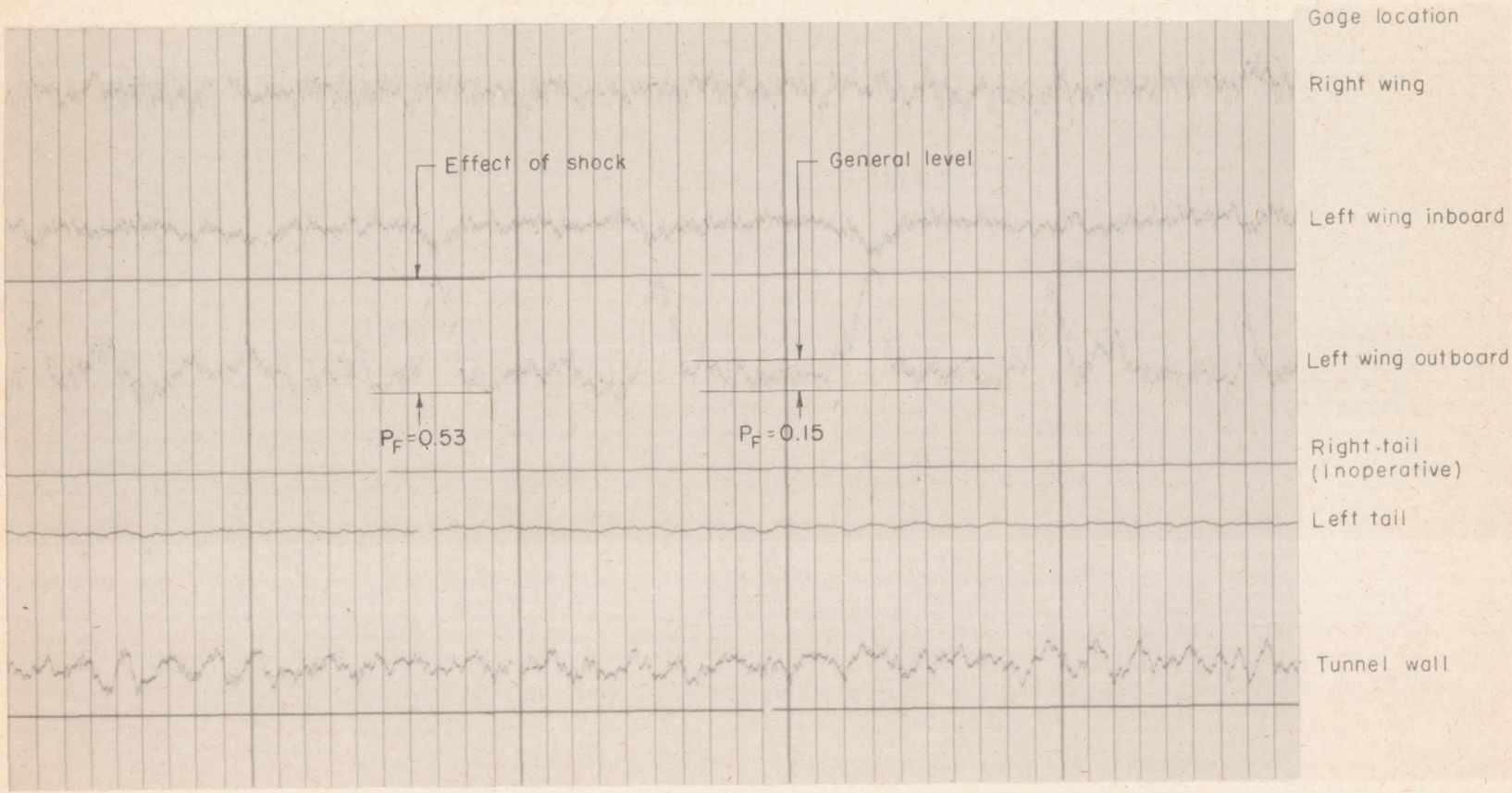


Figure 7.- Portion of oscillograph record indicating shock movement in the vicinity of the outboard gage on the left wing. Mach number, 0.90; model angle of attack, -2.13°.

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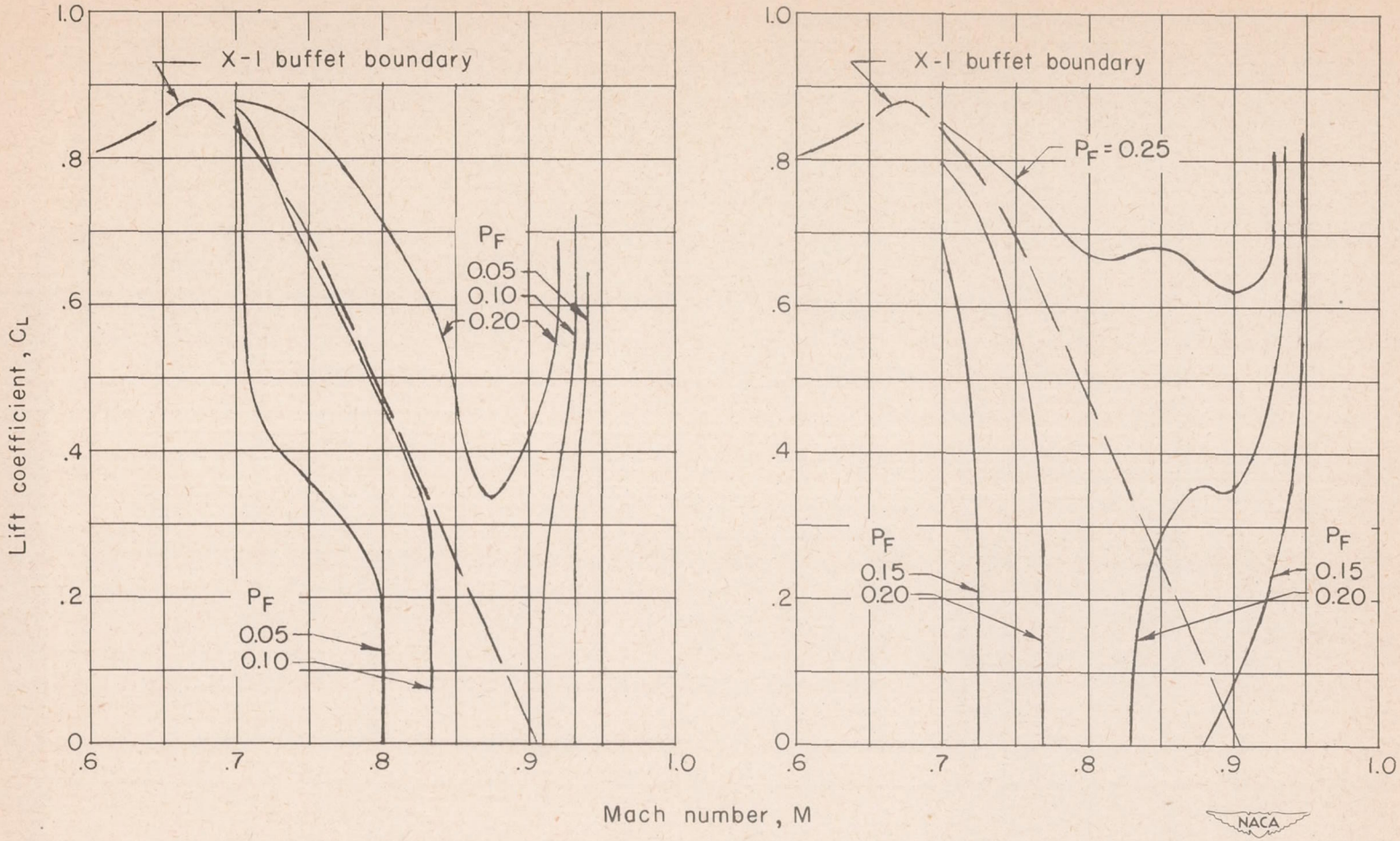


(a) Left-wing-outboard gage.

(b) Left-tail gage.

Figure 8.- Comparison of pressure-fluctuation-coefficient intensities measured on the $\frac{1}{4}$-scale model of the X-1 airplane with the flight-determined buffet boundary of the X-1 airplane.

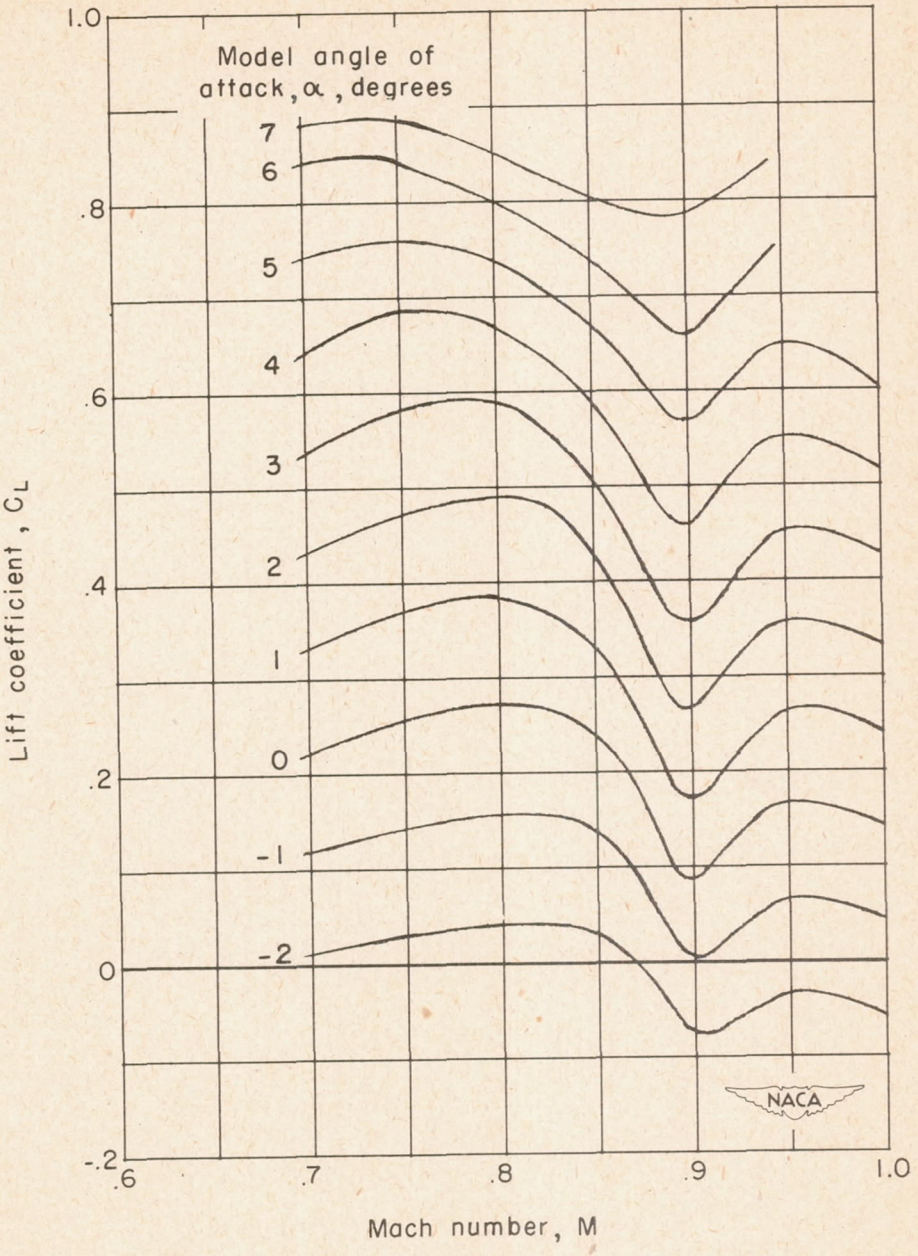
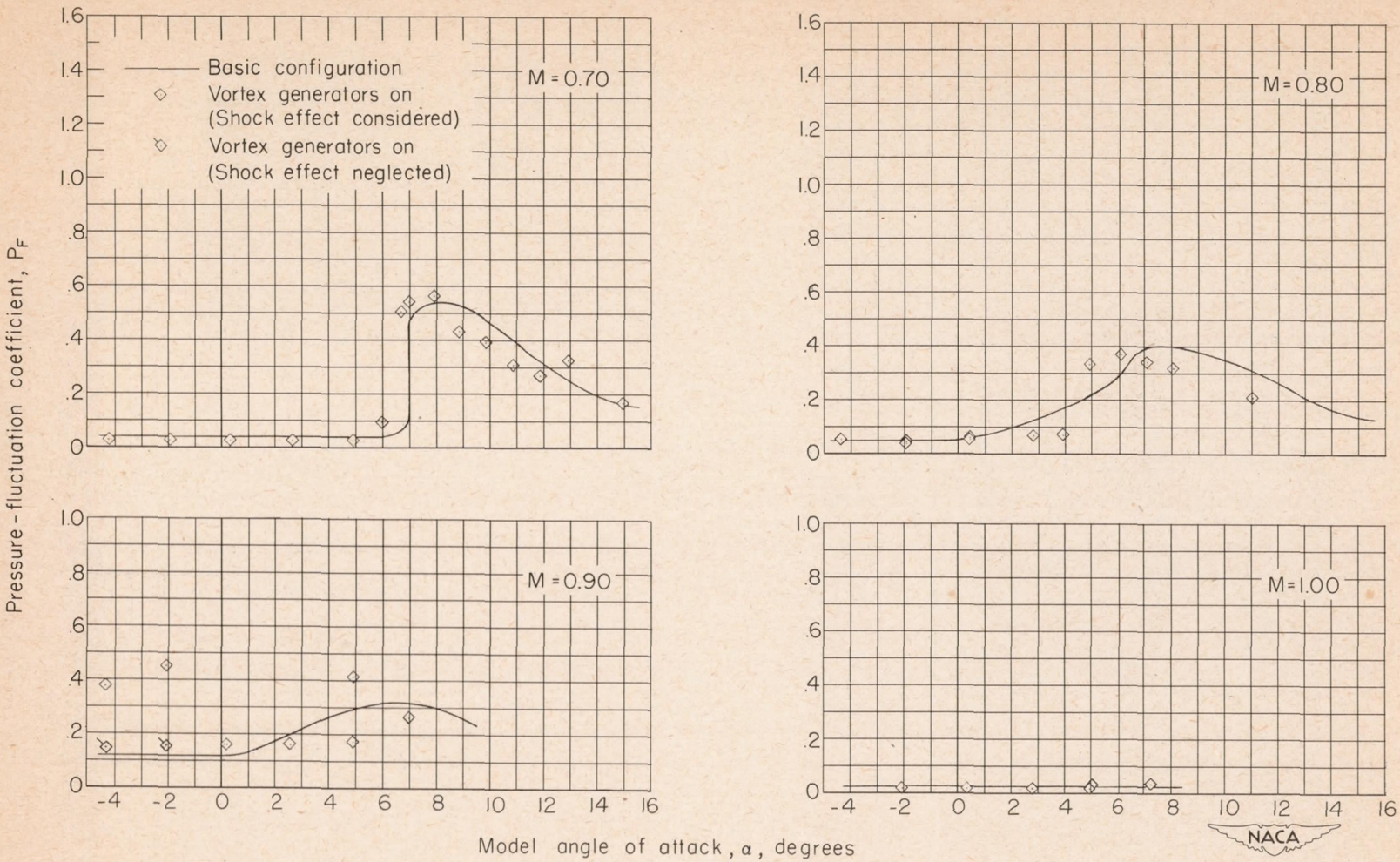
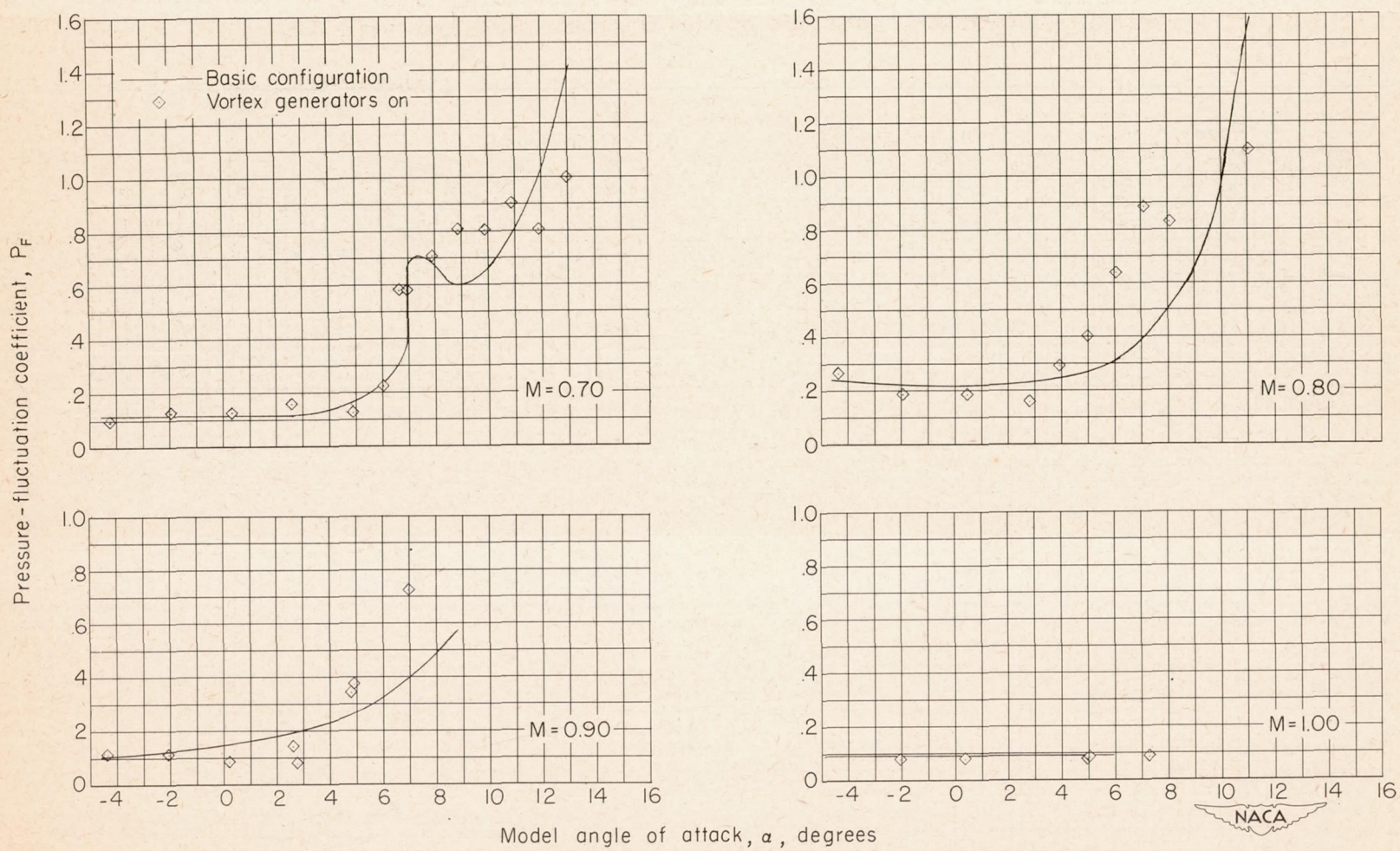


Figure 9.- Variation of lift coefficient with Mach number at various model angles of attack for the $\frac{1}{4}$-scale model of the X-1 airplane in the Langley 16-foot transonic tunnel.



(a) Left-wing-outboard gage.

Figure 10.- Effect of vortex generators on the pressure-fluctuation coefficients.



(b) Left-tail gage.

Figure 10.- Concluded.

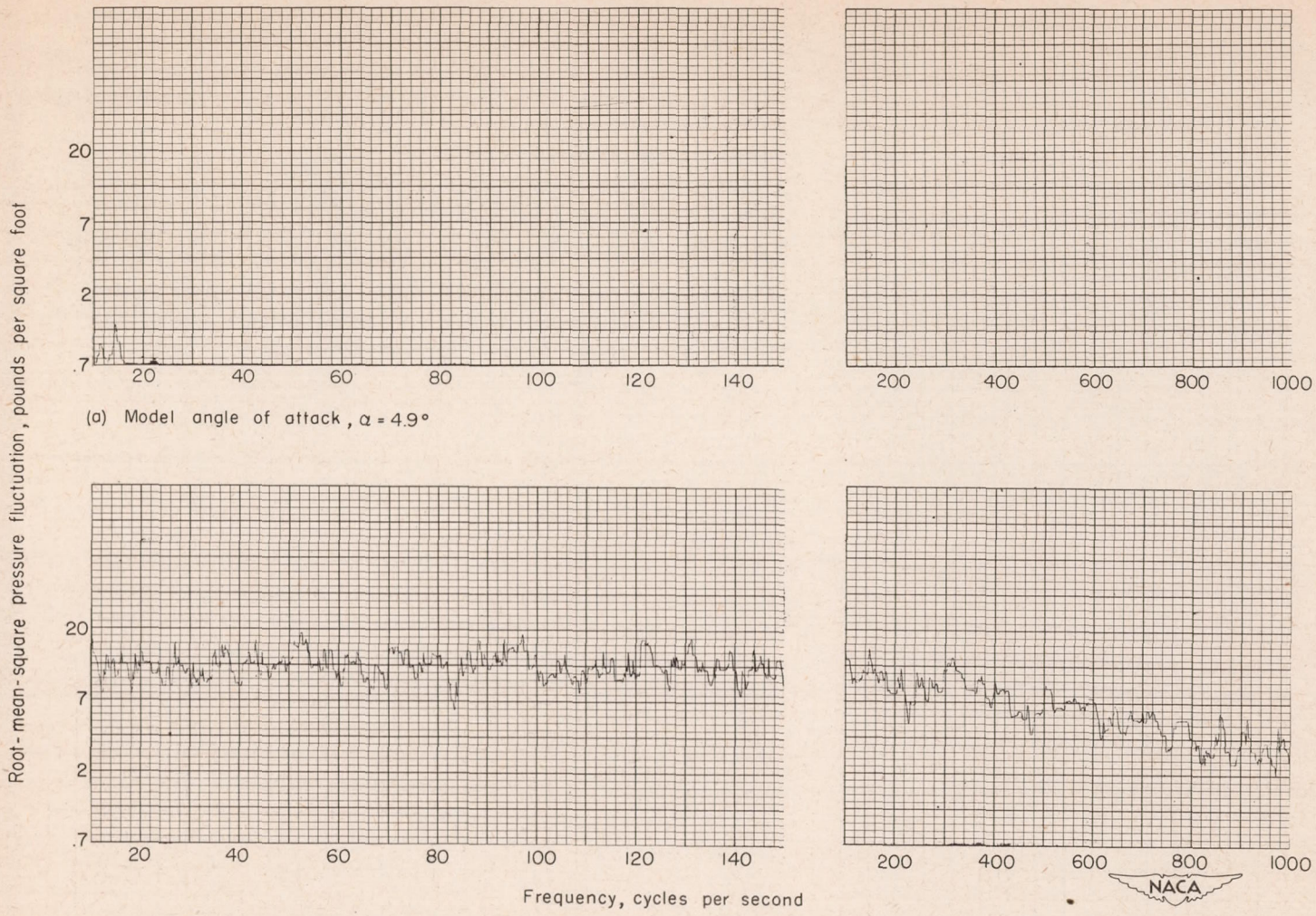


Figure 11.- Frequency analyses of the pressure fluctuations at the left-wing-outboard gage at Mach number of 0.70. Basic configuration.

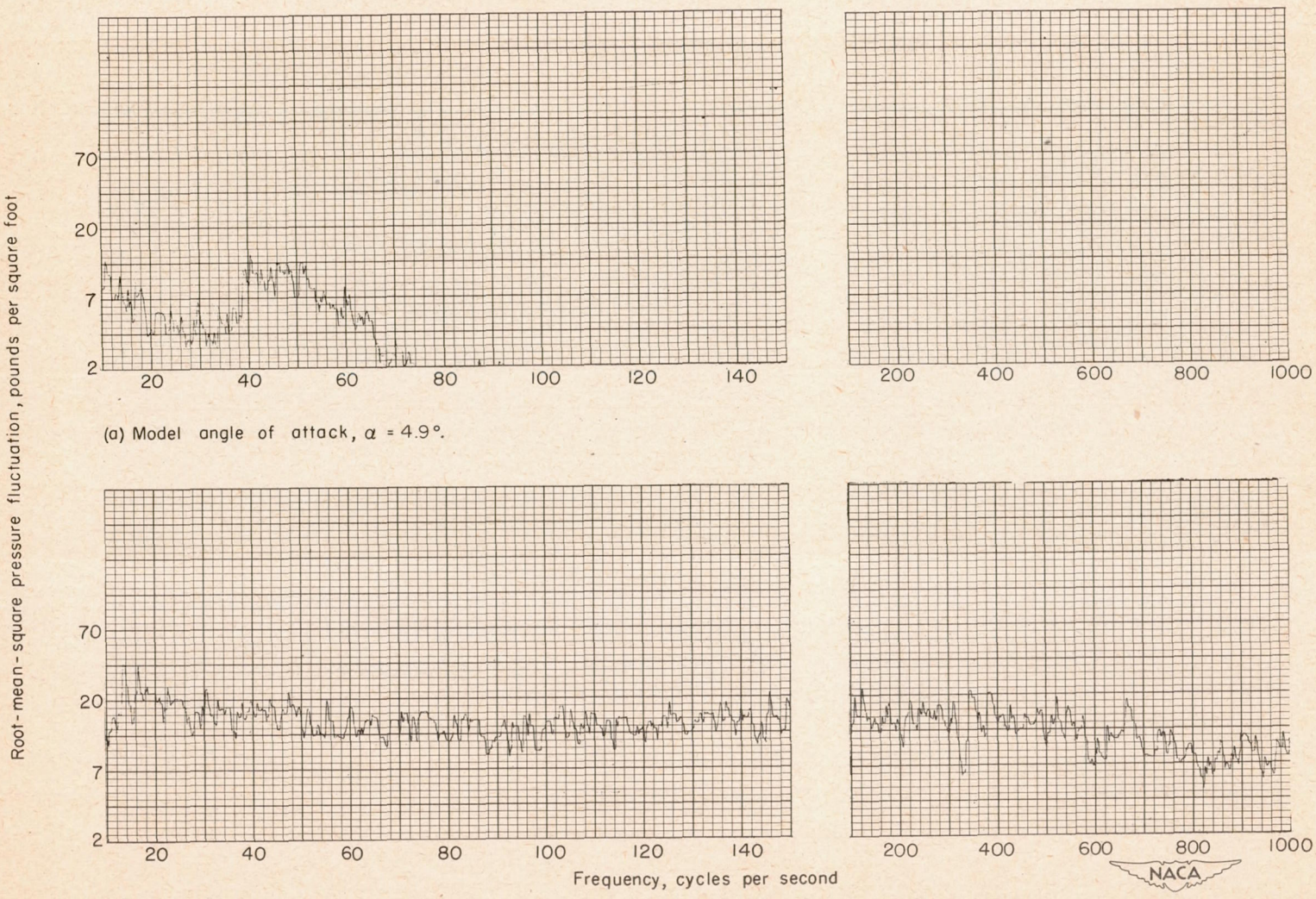


Figure 12.- Frequency analyses of the pressure fluctuations at the left-tail gage at Mach number of 0.70. Basic configuration.

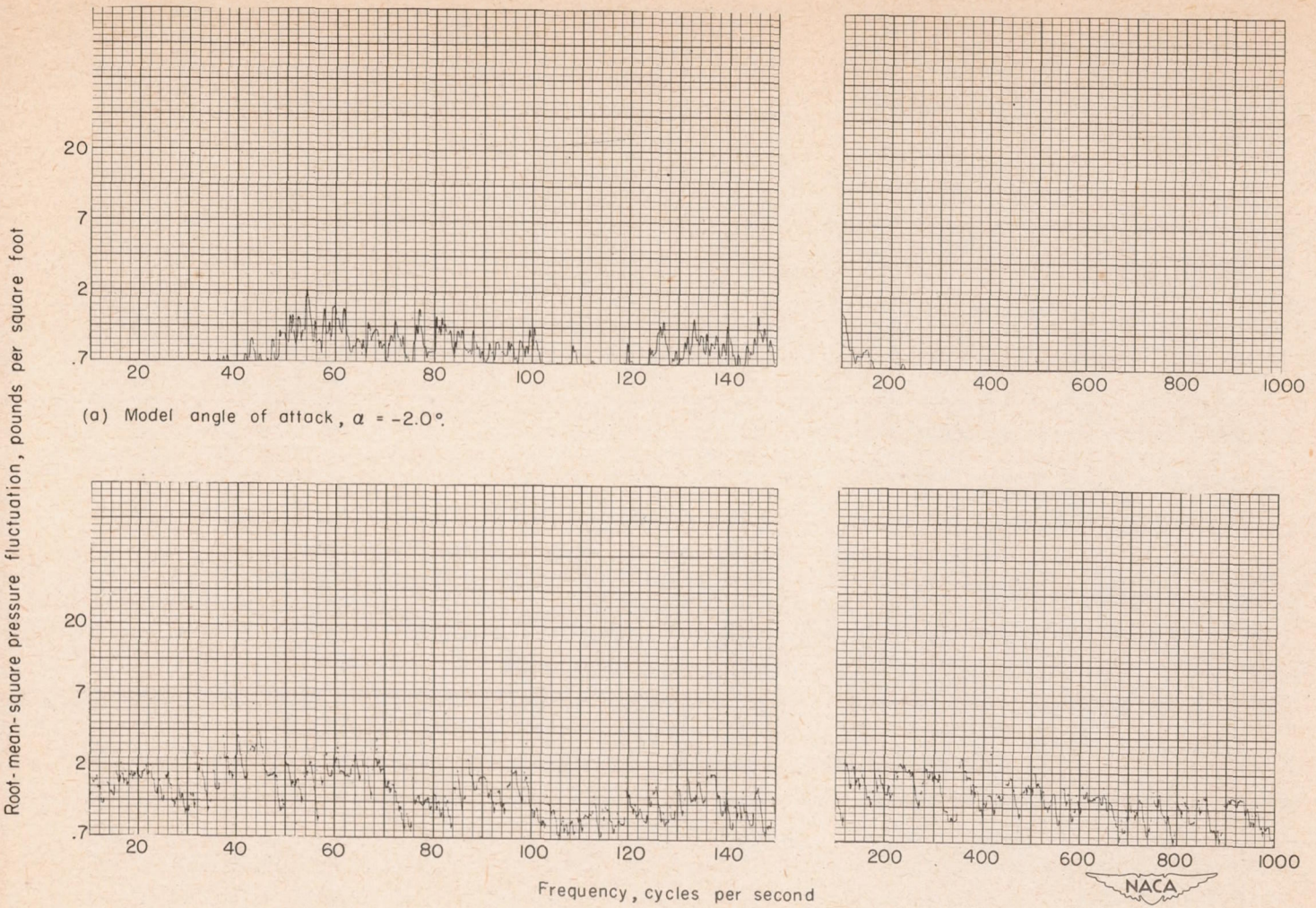


Figure 13.- Frequency analyses of the pressure fluctuations at the left-wing-outboard gage at Mach number of 0.80. Basic configuration.